NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Space Programs Summary No. 37-36, Volume VI

for the period September 1, 1965 to October 31, 1965

Space Exploration Programs and Space Sciences

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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Preface

The *Space Programs Summary* is a six-volume, bimonthly publication that documents the current project activities and supporting research and advanced development efforts conducted or managed by JPL for the NASA space exploration programs. The titles of all volumes of the *Space Programs Summary* are:

Vol. I. The Lunar Program (Confidential)

Vol. II. The Planetary–Interplanetary Program (Confidential)

Vol. III. The Deep Space Network (Unclassified)

Vol. IV. Supporting Research and Advanced Development (Unclassified)

Vol. V. Supporting Research and Advanced Development (Confidential)

Vol. VI. Space Exploration Programs and Space Sciences (Unclassified)

The *Space Programs Summary,* Vol. VI consists of an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of technical supporting activities, including engineering development of environmental-test facilities, and quality assurance and reliability; and a reprint of the space science instrumentation studies of Vols. I and II. This instrumentation work is conducted by the JPL Space Sciences Division and also by individuals of various colleges, universities, and other organizations. All such projects are supported by the Laboratory and are concerned with the development of instruments for use in the NASA space flight programs.

W. H. Pickering

W. H. Pickering, *Director*
Jet Propulsion Laboratory

Space Programs Summary No. 37-36, Vol. VI

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LUNAR PROGRAM

I. *Surveyor* Project

A. Introduction

Calculated to span the gap between the *Ranger* Project and Project *Apollo*, the *Surveyor* Project is designed to take the next step in advancing lunar technology by making soft landings on the Moon with unmanned spacecraft. Various engineering and scientific experiments will be performed during touchdown and while on the lunar surface. The first launches will be engineering test missions to demonstrate system capability up to soft landing and limited postlanding operations. The engineering payload includes elements of redundancy, diagnostic telemetry, touchdown instrumentation, and survey TV.

Following the engineering test missions, the objectives are to extend our knowledge of lunar conditions and to verify the suitability of *Apollo* landing sites. The science payload is planned to consist of two-camera TV, micrometeorite ejecta, single-axis seismometer, alpha particle scattering, soil mechanics surface sampler, and touchdown dynamics experiments.

Hughes Aircraft Company (HAC), Space Systems Division, is under contract to develop and fabricate the first seven spacecraft. The JPL Space Flight Operations Facility and the Deep Space Network (Mission Operations System) will be utilized for flight control and tracking. The launch vehicle will be a combination *Atlas/Centaur*. The first launch is anticipated during the first half of 1966.

B. Systems Testing

1. SC-1 Flight Spacecraft

Solar-thermal-vacuum testing of the SC-1 flight spacecraft was accomplished at HAC during this reporting period. The original test plan called for two complete mission sequences, one at low and the other at high solar intensity, in order to obtain margin performance data. Approximately 12 hr of the first test (an abbreviated version of the total solar-thermal-vacuum test) had elapsed when spacecraft electrical problems necessitated a deviation from the planned test sequence. This first part of the mission had been conducted at an average solar intensity of 82%. It was decided that the remainder of the abbreviated test should be conducted at 100% solar intensity to obtain nominal thermal performance data. Post-test intensity studies indicated that the actual average intensity during the second phase, which lasted 14 hr, was 94 to 95%, with localized intensities of 90 to 91% on most units. Two eclipses (15 and 30 min) and several terminal descent tests were performed during the high-intensity phase.

The test data permitted a preliminary evaluation of the thermal performance of the various spacecraft subsystems. The temperatures were essentially as predicted, and the deviations from specified levels which did occur did not apparently affect subsystem performance. However, various problems experienced during this testing have necessitated a more complete test sequence.

2. SC-2 Flight Spacecraft

The SC-2 spacecraft began initial system checkout at HAC with System Test Equipment Assembly 4. The required power and grounding tests were performed previously. Due to transmitter changes required to support SC-1 spacecraft testing, certain items required retest. The telecommunications intergroup integration test was performed, as well as a radar altimeter and doppler velocity sensor (RADVS) group test. The spacecraft was then shut down to incorporate harness changes to the flight control telemetry circuitry. Following a telecommunications intergroup integration retest, flight control telecommunications testing was begun. Initial system checkout mission sequence, electromagnetic interference, and vernier engine vibration tests are also scheduled.

3. SD-2 Spacecraft Dynamic Model

The SD-2 spacecraft was successfully launched on August 11, 1965, from the Eastern Test Range, using the *Atlas/Centaur 6* launch vehicle (Fig. 1). All flight events occurred at, or very close to, predicted times. The spacecraft and launch vehicle systems performed normally during the launch-to-injection phase. The powered-flight trajectory was close to nominal, and the spacecraft was successfully injected into the simulated lunar transfer trajectory as planned. The injection parameters were well within the established limits, and the SD-2 separation tumble rate was well below the maximum.

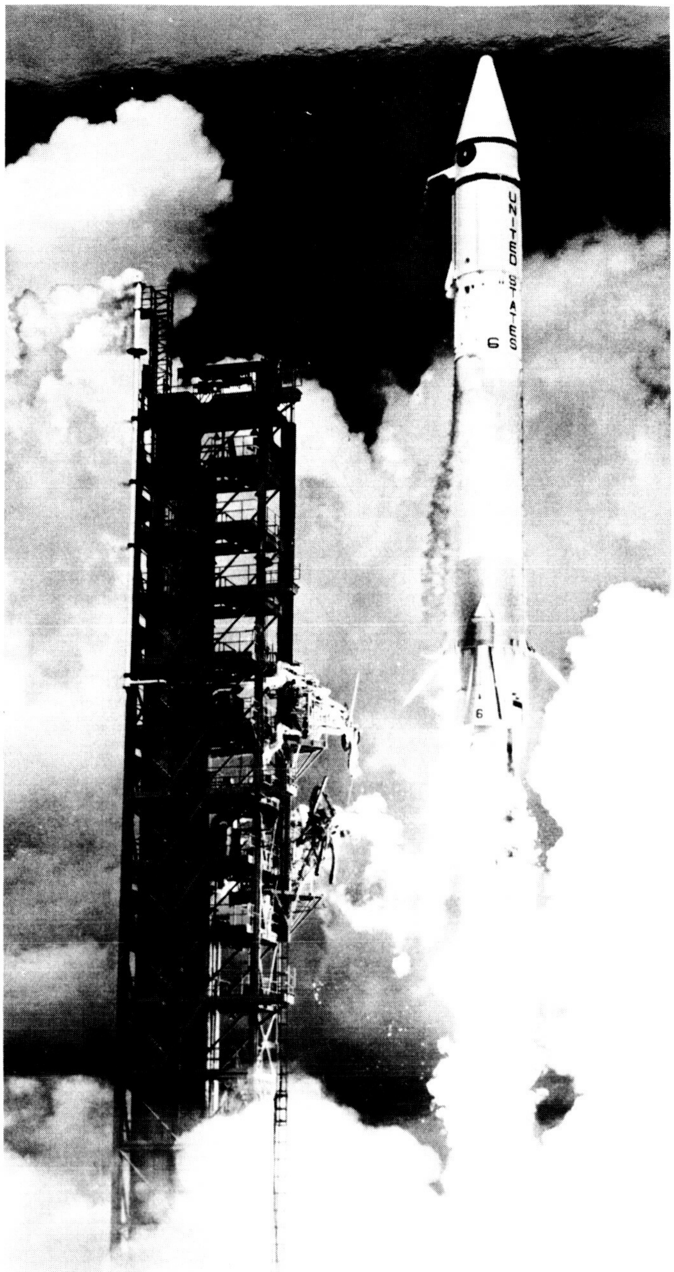

Fig. 1. Launch of the SD-2 spacecraft from the Eastern Test Range

At 16 hr after launch, an abrupt loss of the SD-2 signal occurred. Up to that point, the signal levels compared favorably with predicted levels. An investigation of possible causes for the signal loss was conducted. A large increase in transmitter frequency had been noted during one-way lock, and this increase, if caused by temperature effects, would be indicative of a very high temperature rise. The existence of a thermal-vacuum problem was disclosed during the investigation. The problem was associated with the high-power switching transistors in the electrical conversion unit. These transistors are mounted close to the transmitter voltage-controlled crystal oscillator which controls the transmitter frequency in one-way lock. Appropriate correction action is under way.

The SD-2 launch and flight were successful in meeting program objectives. The mission yielded factual information on the spacecraft physical environment during launch, enabled all tracking stations to obtain actual flight tracking experience, and verified the capability of the *Atlas/Centaur* vehicle to place the *Surveyor* spacecraft in the proper flight path to the Moon. The flight path was such that the simulated landing was only 150 nm from the planned target area.

4. GT-1 Group Test Vehicle

During upgrading operations on the GT-1 vehicle, the new harness, TV, RADVS, and antenna solar-panel positioner were installed. When the upgrade was completed, the SC-2 survey camera and certain SC-3 and SP-1 (flight-quality spare subsystem set) power units were group-tested. After modification, the SC-2 engineering signal processor and auxiliary engineering signal processor were retested in preparation for installation on the spacecraft. Although not physically installed on the GT-1 vehicle, the SC-2 altitude marking radar unit was also group-tested. Group testing of SC-3 and SP-1 units is continuing.

5. T-2N-1 Descent Dynamics Test Vehicle

Two tests of the T-2N-1 vehicle while tethered from a balloon were performed at the Air Force Missile Development Center. A RADVS klystron power-supply modulator failure delayed completion of the first tethered terminal countdown. The malfunction was corrected, and the test was then completed. The second test was completed approximately 2 wk later. Corrective action was taken on all problem areas. Acoustic isolation between the vernier engines and the RADVS subsystem was inadequate for both tests, but all subsystems except RADVS exhibited satisfactory performance in the engine noise environment. Pre-release checkout of all subsystems was satisfactory, except for the RADVS test fixture jettison for the second test. Satisfactory countdown procedures, abort sequence, and postmission downloading; compatibility of the operation display system and crew (except for recurrent communications system problems); adequacy of ground control facilities and ground handling procedures; and pitch, yaw, and roll stability were demonstrated.

6. T-2N-2 Descent Dynamics Test Vehicle

System functional environmental testing of the T-2N-2 vehicle was satisfactorily completed. Vehicle flight-acceptance testing, consisting of RF interference, static discharge, vibration, thermal and humidity, and landing shock phases, was then performed. When the RADVS system is returned from the manufacturer's facility where it is being upgraded, RADVS functional tests will be performed, and the T-2N-2 vehicle will then undergo one additional vibration phase before shipment to the Air Force Missile Development Center.

7. T-2N-R Recovery System Test Vehicle

The T-2N-R drop-test program was satisfactorily completed during this reporting period. Due to high pitch, yaw, and roll moments exhibited during the first four tests, the aerodynamic balance plate area was increased. Since the pitch and roll moments were still marginal during the next two tests, the area of the plate was again increased, and three airfoils were added. Acceptable aerodynamic moments were then demonstrated in a following test. The recovery electronics assembly, telemetry conditioning, and solenoid release mechanism and safety lock were all found to operate satisfactorily. In-tolerance parachute deployment time and effective recovery distance; satisfactory descent and touchdown stability; adequacy of ground control facilities and ground handling equipment; in-tolerance force with no structural damage during parachute deployment; satisfactory performance of the recovery system; and in-tolerance average sink rates were demonstrated.

8. T-21 Prototype System Test Spacecraft

The T-21 spacecraft was shipped to the Goldstone Pioneer Station after completing the Pathfinder Mission (*SPS 37-35*, Vol. VI, p. 2) at the Eastern Test Range. Preparation of the spacecraft for testing consisted primarily of a performance verification test, a system readiness test, and special operations to render the spacecraft compatible with Deep Space Instrumentation Facility (DSIF) test requirements. Considerable difficulty in obtaining repeatable measurements of spacecraft/DSIF RF-link losses early in the testing necessitated the replacement of one of the two parabolic antennas on the collimating tower with a larger antenna. Following this corrective action, RF, command, telemetry, TV, and space flight operations compatibility tests were performed. Also, various spacecraft malfunctions unannounced to the DSIF operations personnel were simulated in order to exercise these personnel in the analysis of spacecraft failure symptoms. The T-21 System Test Equipment Assembly 3 performed satisfactorily throughout the tests, with only minor perturbations reported. The T-21 spacecraft was then scheduled for shipment to HAC for preparation for drop testing.

C. Flight Control

A production model of the new Canopus sensor shade design incorporating a greater number of baffles than previously used was subjected to stray light and moonlight interference tests such as were performed on the developmental model (*SPS 37-34*, Vol. VI, p. 16). A comparison of the production model data and the developmental model data revealed no significant differences in performance. The susceptibility of the Canopus sensor to free particles that may enter the sensor field of view was also investigated. No significant degradation in performance is expected to result from this source.

An operating roll actuator was installed on the S-7 test spaceframe in order to exercise the roll actuator and Vernier Engine 1 during the type-approval vibration testing of the vernier propulsion system at the JPL Edwards Test Station. The roll actuator was commanded through a series of plus and minus roll positions during simulated midcourse and vernier descent firings. Preliminary test results indicate that all roll actuator responses were satisfactory and no failures occurred.

D. Electronics

1. Transmitters

Efforts to make the *Surveyor* transmitters insensitive to the critical-air-pressure region are continuing. Although the application of high-density foam provides a temporary solution, a more acceptable solution is still being sought. Test results show that the compartment is not well-vented, as evidenced by an internal pressure lag over external pressure; therefore, the compartment will be at critical pressure at the time of high-power turn-on (approximately 11 min after launch). Since the transmitter contains high voltage in the electrical conversion unit and high RF voltage when operated through the diplexer, the following modifications were found necessary: (1) rerouting of high-voltage wires in the electrical conversion unit, (2) foaming of high-voltage areas, and (3) arc-proofing of the diplexer so that this unit will not be subject to the pressure lag. Three transmitters upgraded to the high-voltage configuration have passed portions of the critical-pressure high-voltage test.

2. Low-Gain Upper-Hemispherical Antenna

It was previously reported (*SPS 37-35*, Vol. VI, p. 3) that consideration was being given to the addition of another antenna on the *Surveyor* spacecraft to improve the over-all performance of the telecommunications subsystem. Since that time, it was decided to suspend efforts toward the development of this antenna due to the potential weight and casting problems which would result from its incorporation.

E. Electrical Power Supply

Special tests were conducted on the boost regulator, battery charge regulator, and engineering mechanism auxiliary to verify performance and to increase the confidence level in these units. Margin testing was performed on the battery regulator and battery charge regulator in the unit areas to determine any malfunctions that may occur as a result of a worst-case analysis. This testing consisted of a thermal-vacuum test to assure dissipation capability and to observe any arc-overs as the pressure was reduced. Another portion of the battery-charge-regulator margin testing checked maximum power transfer over the thermal range and battery voltage range. All units passed the tests, except for one battery charge regulator which dropped in efficiency when operating at conditions not normally encountered in the compartment.

F. Propulsion

1. Vernier-Propulsion-Subsystem Component Evaluation

Work continued at JPL on the program to perform limited environmental tests of selected upstream system components for the purpose of evaluating, on an independent basis, the capability of these components to meet the *Surveyor* requirements. The second component to be evaluated was the helium relief valve, which prevents overpressurization of the propellant tanks in the event of a malfunction during regulator lockup.

Early in the development of the valve, design problems were encountered in the sealing of the valve at low temperatures. A Teflon seal which was incorporated caused difficulty in assembly, since the seal had to be stretched during installation on the piston. Since this deformed the seal and resulted in excessive leakage, it was decided to change the piston design to eliminate the need for stretching of the seal.

For this evaluation, two valves were acquired from HAC: one of the original design and the other in the modified configuration with the redesigned piston. Due to a structural fatigue failure, testing on the original-design valve was discontinued after the high-temperature cycling test.

Prior to its evaluation, the modified valve was cycled 1000 times, using the original piston to "wear-in" the Teflon seal which had a leakage greater than the specification value. After this cycling, the valve passed all leakage tests with the exception of the post-vibration test, where the leakage was greater than that allowed by the specification. Since subsequent leak checks were within tolerance, it was suspected that this failure was caused by some particulate matter in the test system that prevented complete closing of the valve. From the results of the testing, it appears that the modified two-piece piston has eliminated the piston-seal leakage problem.

Satisfactory cracking and reseat pressures, valve response, and valve capacity were also demonstrated during the tests. As an additional test, the modified valve will be given an exposure to oxidizer vapors to verify propellant compatibility.

2. *Vernier-Propulsion-Subsystem Type-Approval Testing*

The thermal environment phase of the vernier-propulsion-subsystem type-approval test program was successfully completed at the HAC Placerita Canyon test facility. The S-6 test spaceframe underwent a real-time simulated mission at each of two temperature extremes. Each simulated mission consisted of a 50-sec midcourse firing and a programmed 175-sec descent firing. The quality of the data recovered was excellent, and all program objectives were achieved.

The vibration environment phase of the type-approval test program was also completed during this reporting period. The S-7 test spaceframe used in these tests underwent vibration at flight-acceptance-test levels two times in each of three orthogonal axes and was then subjected to vibration at type-approval-test levels in both horizontal axes. After these tests, which were conducted at the HAC Space Simulation Laboratory, the vehicle was transferred to the JPL Edwards Test Station for simulated boost vibration at type-approval-test levels, two simulated midcourse firings without vibration, and a simulated descent firing with retromotor vibration. Data from these tests are currently being analyzed.

3. *Thrust Chamber Assembly Evaluation*

The evaluation of *Surveyor* vernier engines continued at the JPL Edwards Test Station. Two tests were performed with a thrust chamber assembly to determine any change in maximum thrust which might result from helium dissolved in the propellants. The data indicate that dissolved helium does not appreciably affect the maximum thrust capability of the engine.

Additionally, tests were performed to demonstrate start characteristics in a cold ultrahigh-vacuum environment. A special vacuum chamber was developed for maintaining the ultrahigh vacuum on the assembly (mounted within the chamber) during the ignition phase. Ignition lag and start transient characteristics during the first test, with the propellants conditioned to about 50°F, were comparable to those demonstrated by the flight-type engine used in previous tests. The second test was performed with propellants conditioned to approximately 20°F. The ignition lag during this test was about 15 to 20 msec longer than that in the first test. Considerable difficulty was experienced during the tests in maintaining a completely leak-tight system with the cold propellants in the ultrahigh-vacuum environment. A third and final test with the propellants conditioned to 0°F is planned.

PLANETARY–INTERPLANETARY PROGRAM

II. *Mariner* Project

A. Introduction

The early objective of the Planetary–Interplanetary Program was the initial probing of the planets Mars and Venus by unmanned spacecraft. The initial probing of Venus was successfully accomplished by the *Mariner II* spacecraft in 1962. The initial probing of Mars was successfully accomplished by the *Mariner IV* spacecraft during the 1964–1965 flight opportunity.

The primary objective of the *Mariner C* missions (*Mariner* Mars 1964 Project) was to conduct closeup flyby scientific observations of the planet Mars during the 1964–1965 flight opportunity and to transmit the results of these observations back to Earth. A secondary objective of the *Mariner C* missions was to provide experience and knowledge about the performance of the basic engineering equipment of an attitude-stabilized flyby spacecraft during a long-duration flight in space farther away from the Sun than the Earth. An additional objective was to perform certain field and particle measurements in interplanetary space during the trip in addition to those performed in the vicinity of Mars.

The *Mariner III* spacecraft was launched from the Eastern Test Range on November 5, 1964. The nose cone failed to eject from the spacecraft, thereby precluding solar-panel deployment. The battery power was depleted 8 hr, 43 min after launch.

The *Mariner IV* spacecraft was successfully injected into a Mars encounter orbit about the Sun on November 28, 1964. The encounter with Mars, which occurred on July 14 and 15, 1965, was entirely successful. Flight operations were conducted as planned with a series of ground-transmitted commands. Planetary observations were performed which provided basic new information about Mars, including TV pictures, cosmic dust particle counts, and measurements of magnetic and ionizing radiation fields. In addition, an Earth occultation experiment was carried out during the Type I trajectory flyby to obtain data relating to the scale height and pressure in the atmosphere of Mars. Analysis of the TV pictures and the data from the other science experiments is continuing.

Phase 1 of the *Mariner IV* mission was successfully concluded October 1, when a command was sent by the Goldstone Venus Tracking Station to transfer the spacecraft transmitter from the high-gain antenna to the omni- or low-gain antenna. The Deep Space Network was in continuous contact, with the spacecraft (with minor exceptions) for the entire 307 days of Phase 1. The total arc distance travelled by *Mariner IV* during that period was 418 million miles. A total of 85 commands were transmitted, successfully received, and acted upon by the spacecraft, and approximately 50 million bits of data were received from the spacecraft. The only anomalies which occurred were degradation of the solar plasma probe data and failure of the Geiger–Müller 10311 tube and the ionization chamber experiment.

The conclusion of Phase 1 marked the beginning of Phase 2, during which the Deep Space Network will attempt to track *Mariner IV* as an RF source completely around its orbit of the Sun. These attempts will be made approximately once every month; however, the earliest significant efforts will probably not take place until the new 210-ft-diameter antenna at the Goldstone Mars Tracking Station is in a research-and-development operational mode (approximately spring of 1966).

B. *Mariner IV* Space Flight Operations

The postencounter events of the *Mariner IV* mission are given in Table 1. During the playback of the second recording sequence beginning September 1, the camera "saw" dark space and stepped through its four gain levels sequentially on the first five recorded pictures. This operation was performed to provide additional data on the dark response of the camera system to aid in the interpretation of the haze or fogging of the *Mariner IV* pictures. After the playback was completed, operations were directed toward returning the spacecraft to a cruise condition. During the remainder of September, the interferometer effect precluded the transmission of any more commands. This condition existed until just prior to the completion of Phase 1 on October 1.

C. *Mariner IV* Power Subsystem Performance

With the exception of the higher-than-expected battery voltage (*SPS 37-33*, Vol. VI, p. 18; and *SPS 37-34*, Vol. VI, p. 20), the operation of the *Mariner IV* power subsystem was completely normal and predictable throughout the 307 days of Phase 1. (After the antenna changeover on October 1, no further data could be received.) Two theories have been advanced to explain the increase in battery voltage:

(1) It was a normal consequence of the 0-*g* gravitational field, temperature effects, and the small (1.6-ma) charge current produced by the battery voltage telemetry transducer.

(2) The battery case had cracked and electrolyte was leaking out slowly and evaporating; this produced a partially open cell, which the battery voltage telemetry transducer interpreted as increased voltage.

Table 1. Postencounter events of the *Mariner IV* mission

Event	Date, 1965	GMT, hr:min:sec
Maneuver inhibit command	August 26	21:06:52
Command for minimum pitch turn	August 26	21:15:16
Command for minimum roll turn	August 26	21:23:40
Command for minimum motor burn	August 26	21:32:04
Canopus angle update command	August 27	19:40:00
Command for turn-off of tape recorder power and start of scan platform	August 30	20:30:00
Command for telemetry Data Mode 3 (all science data)	August 30	21:10:24
Command for inhibit of scan platform	August 30	22:48:38
Command for start of record sequence	August 30	23:35:26
Command for telemetry Data Mode 2	August 31	00:05:00
Command for turn-off of all science and turn-off of battery charger	August 31	00:44:00
Command for switching tracks on tape recorder	August 31	00:49:00
Command for telemetry Data Mode 4 playback	August 31	01:25:00
Playback of second recording sequence	September 1 to September 2	02:00:46 to 06:48:32
Command for turn-off of video system and turn-on of battery charger	September 2	06:17:00
Command for turn-off of battery charger	September 2	06:23:00
Command to place spacecraft in cruise mode	September 2	06:29:00
Command to switch spacecraft from high-gain to low-gain antenna	October 1	21:30:17
Receipt of last signal from spacecraft	October 1	22:05:07

When the antenna changeover was made, the battery voltage had increased to 37.2 v dc. Since the voltage did not exceed 38.0 v dc – a limit at which the second theory would become more likely – a positive determination of which theory is correct does not appear possible. Numerous tests with identical batteries and with individual cells and conferences with the battery manufacturer did not yield conclusive results. Since a definite solution using *Mariner IV* data seems unlikely, it is hoped that continued study of silver-zinc batteries may provide the answer.

III. *Voyager* Project

The primary objective of the *Voyager* Project is to conduct scientific investigations of the solar system by instrumented, unmanned spacecraft which will fly by, orbit, and/or land on the planets. Emphasis will be placed on the acquisition of scientific information relevant to the origin and evolution of the solar system, the origin, evolution, and nature of life, and the application of this information to an understanding of terrestrial life.

The primary objective of the portion of the *Voyager* Project relating to the investigation of Mars is to obtain information relative to the existence and nature of extraterrestrial life and relative to the atmosphere, surface, and body characteristics of Mars and the planetary environment by performing a series of experiments on the surface of, and in orbit about, the planet. For the first mission, which is presently scheduled for 1971, primary emphasis will be placed on an orbiter, with the possible inclusion of a nonsurviving, engineering test capsule or atmospheric-entry probe. The first capsule missions to include scientific experiments on the surface of Mars are currently scheduled for the 1973 opportunity.

A secondary objective is to further our knowledge of the interplanetary medium between the planets Earth and Mars by performing scientific and engineering measurements while the spacecraft is in transit.

Two *Voyager* planetary vehicles are to be designed, constructed, and tested for launch during the Mars 1971 opportunity. A single *Saturn* V is planned to serve as the launch vehicle for this mission. Attention is also being given to requirements imposed on the vehicle by similar launches subsequent to 1971. The vehicle design will be such as to enable large scientific payloads to be carried to the planet, a large amount of data to be telemetered back to Earth, and long useful lifetimes in orbit about the planet and/or on the planetary surface. Hardware will be designed to accommodate a variety of spacecraft and/or capsule payloads, mission profiles, and trajectories. Particular emphasis will be given to simple and conservative design, redundancy wherever appropriate, and a comprehensive program of component, subsystem, and system testing. No deep-space flight tests of the spacecraft are planned; however, the test program for the capsule may include Earth-entry flight tests using an *Atlas* or smaller launch vehicle.

Present design plans for the *Voyager* spacecraft are as follows: The flight spacecraft will be fully attitude-stabilized, utilizing celestial references for the cruise phase. The capability of providing velocity increments for midcourse trajectory corrections and for Mars orbit attainment by both the spacecraft and capsule will be provided. Onboard sequencing and logic and ground command capability will also be included. The spacecraft, thermally integrated and stabilized, will supply its own electrical power from either solar energy or internal sources and will be capable of maintaining radio communications with Earth. Data on various scientific phenomena near Mars and data on spacecraft performance will be telemetered back to Earth.

The flight spacecraft will provide the capsule with services such as electrical power, timing and sequencing, telemetry, and command during the transit portion of the missions, and it may also serve as a communications relay. The sterilized capsule will be designed to separate from the orbiting spacecraft, attain a Mars trajectory, enter the Martian atmosphere, descend to the surface, and (for the 1973 mission) withstand impact and attain surface lifetimes of as much as 6 months.

Mission engineering studies were undertaken to derive a model sequence of *Voyager* missions over several planetary opportunities to serve as the basis for 1971 *Voyager* mission planning. On the basis of the 1971 *Voyager* mission specification, spacecraft system analyses and tradeoff studies are being performed by JPL and three spacecraft contractors to assess the constraints imposed on the spacecraft by the mission objectives and by other systems. A single spacecraft contractor will be selected after completion of preliminary designs. This contractor, under JPL technical direction, will then be responsible for:

(1) Refining the preliminary designs and functional descriptions.

(2) Developing system and subsystem functional specifications.

(3) Breadboarding and testing selected subsystem hardware.

(4) Preparing, for JPL approval, detailed plans covering the proposed management and implementation of the development, fabrication, testing, launch, and test operations efforts.

Study efforts in areas such as guidance and control, telecommunications, and propulsion are continuing at JPL.

IV. Future Projects

A. Introduction

The JPL Future Projects Office is responsible for the direction, sponsorship, and, in cooperation with other technical areas of the Laboratory, origination of advanced mission and future project studies of interest and applicability to the Laboratory space program. The objective is to ensure the availability, when required, of preliminary technical information and plans for potential missions, including associated information on spacecraft systems, instrumentation, launch vehicle systems, resources, and schedule requirements. This preliminary information will be used to define specific flight mission projects and to provide general guidelines for the direction of Laboratory research and advanced development efforts. A summary or informative abstract of the results of each study will be presented in the *Space Programs Summary*, including, where appropriate, reference to the more detailed formal final report issued by the Future Projects Office.

B. A Systems Comparison of Direct- and Relay-Link Communications for an Eventual Long-Life Mars Surface Experiment[1]

This article summarizes the results of an evaluation of two alternate schemes for returning data to Earth from an eventual long-life Mars surface experiment. One scheme involves transmission of the lander data to an artificial planetary satellite for subsequent transmission to Earth, and the other involves transmission directly from the lander to Earth. The relay system, requiring the

[1]A detailed description of this analysis is presented by Barber, T. A., Billy, J. M., and Bourke, R. D., in *A Systems Comparison of Direct and Relay Link Data Return Modes for Advanced Planetary Mission*, Technical Memorandum No. 33-228, Jet Propulsion Laboratory, Pasadena, California (To be published).

orbiting and simultaneous operation of a communications satellite with the landing of the capsule, represents a considerable increase in complexity over the direct-link system. The satellite receives data from the lander sporadically at a high rate and transmits it continuously to Earth at a low rate. With the direct system, on the other hand, it is necessary to erect and steer a highly directional antenna to support the required data rate for reasonable powers. The requirements for these operations serve to degrade the reliability of the direct-link system. Furthermore, since the lander is on the surface of a rotating planet, it can view and transmit to Earth during only a fraction of the day. Both systems must return the data over the enormous, continuously changing Earth–Mars range. These and other technical aspects of the two schemes were evaluated so that their data return capabilities could be quantitatively compared.

Three comparison criteria were used:

(1) Effective information rate $\dot{I}_e = \dot{I}_i \cdot v$, where $\dot{I}_i$ is the actual information rate when communications are established, and v is the fraction of time that transmission is actually carried out. Whereas the direct scheme has only one effective information rate, the relay scheme has two: one for the lander-to-orbiter link and the other for the orbiter-to-Earth link. The smaller of these two determines the amount of lander data received at Earth.

(2) Reliability $R(t)$, the probability that each system is operating properly at time t. This has been obtained through reliability and failure-rate models of subsystems and required functional sequences.

(3) Expected total bits

$$I(T) = \dot{I}_e \int_{t_0}^{T} R(t)\, dt,$$

where t_0 is the time at which data begin to accrue, and T is greater than t_0 and less than or equal to the end-of-mission time. This quantity is actually the measure of the average amount of information received at Earth using one particular scheme.

To numerically evaluate these three comparison criteria, two basic assumptions were made: (1) The lander was assumed to gather and return to Earth 10^{10} bits of data over a 6-mo period, and (2) the opportunities of 1975 and 1977 were assumed for trajectory purposes. As a consequence of the first assumption, the following telecommunications system characteristics were used in this analysis: The Mars-to-Earth links for both schemes were taken to use steerable high-gain (~35-db) 12.5-ft-diameter antennas, 50-w S-band transmitters, Earth coverage as required with the 210-ft-diameter Deep Space Instrumentation Facility antennas, command playback, and adequate on-board data storage. The lander-to-orbiter link of the relay system was taken to use a VHF transmitter; an antenna on the planet surface erected to the local vertical; an antenna on the orbiter directed toward the planet center, which, in conjunction with the lander antenna, was to give about 10 db of total gain; command playback; and automatic acquisition. Adequate storage (a few percent of the total bits) to prevent the loss of significant amounts of data was assumed for both ends of the link.

The results of this study, shown in Table 1, may be summarized as follows:

(1) For the assumed parameters, the effective bit rate of the relay link is better by a factor of three.

(2) The relay link is less reliable than the direct link, being about twice as likely to fail during the 6-mo mission.

(3) The amount of expected cumulative data using the relay link is greater than that using the direct link by a factor of two.

(4) For the assumed lander relay transmitter power, a 4000- × 20,000-km orbit reduces the relay-link effective bit rate below that of the direct link.

Table 1. Results of the direct- and relay-link systems comparison

Comparison criteria	Relay system: Lander-to-orbiter link[a] with indicated orbital altitude: 5000 km	Relay system: Lander-to-orbiter link[a] with indicated orbital altitude: 2000 km	Relay system: Lander-to-orbiter link[a] with indicated orbital altitude: 4000 × 20,000 km	Relay system: Mars-to-Earth link[b]	Direct Mars-to-Earth link
Viewing fraction v	0.118	0.038	0.124	~1	0.33
Actual information rate $\dot{I}_i$, bits/sec	28,300	143,000	2930	3340	3340
Effective information rate $\dot{I}_e$, bits/sec	3340	5450	364	3340	1110
Total bits (perfect reliability), × 10^{10}	5.2	8.5	0.57	5.2	1.7
Expected total bits $I(T)$, × 10^{10}	0.63	1.03	0.069	0.63[c]	0.33[c]

[a] 10-w lander transmitter; 10-db antenna system gain.
[b] 35-w radiated power; 36.7-db antenna system gain.
[c] 400-day transit; 0.75 direct-link antenna reliability.

(5) Both systems are data-rate-limited by the Earth link.

(6) The 10^{10}-bit requirement necessitates the use of a high-power transmitter and a very-high-gain steerable antenna for both systems.

On the basis of these results, it may be concluded that both systems yield comparable expected total data return and that, consequently, the choice between a direct- and relay-link system should be made on grounds other than data return capability.

DEEP SPACE NETWORK

V. Deep Space Network Systems

A. Introduction

The Deep Space Network (DSN) is the NASA precision communications network designed to communicate with, and permit control of, spacecraft designed for deep space exploration. The DSN consists of the Deep Space Instrumentation Facility (DSIF), the Space Flight Operations Facility (SFOF), and the DSN Ground Communications System (GCS).

It is the policy of the DSN to continuously conduct research and development of new components and systems and to engineer them into the DSN to maintain a state-of-the-art capability. Presently, a capability exists for simultaneous control of a newly launched spacecraft and a second one already in flight. Efforts are under way to provide for simultaneous control of either two newly launched spacecraft plus two in flight or four spacecraft in flight.

B. DSN Monitoring System

1. Introduction

A DSN Monitoring System is to be designed and developed to provide monitoring at points along the data flow path so that a measure of DSN performance in real or near-real time may be obtained. A data-monitoring function and performance and failure data in real or near-real time for information purposes, for use in taking corrective action, and to aid in validation of output data will be provided, as well as a permanent record of performance and data monitoring. Functional specifications are being reviewed and updated. The design, development, installation, and checkout of the DSN Monitoring System for support of the *Surveyor* and *Lunar Orbiter* Projects are scheduled for completion in spring 1967.

2. DSN Monitor Area for the SFOF

The DSN Monitor Area, functionally a part of the over-all DSN Monitoring System, will be the primary user subsystem of the monitoring information sensed, processed, and provided by the System. As such, it will be a mission-independent, highly reliable, on-line facility located in the SFOF. Selected outputs from the DSN Monitoring System and SFOF subsystems will be displayed to aid in the validation of DSN output data. The information flow and control element interfaces are shown in Fig. 1. The area will house two DSN Monitor and Analysis teams and appropriate input/output and display equipment. Sufficient capacity for simultaneous support of two missions (two stations per mission) will be provided. A functional specification for the SFOF displays and the display buffer is being prepared, and an operational design team for the DSN Monitor Area has been formed.

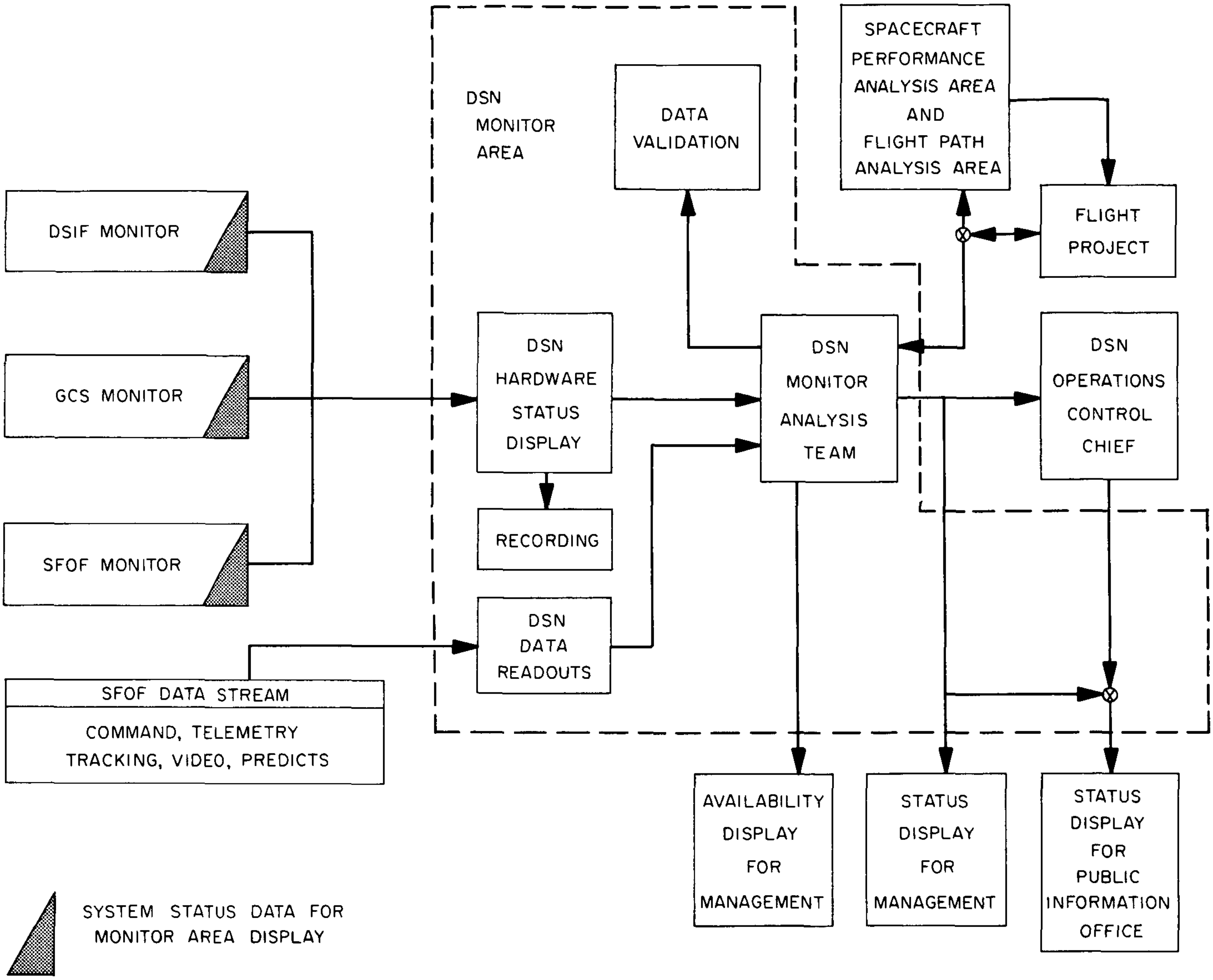

Fig. 1. DSN Monitor Area information flow and control element interfaces

C. DSIF Chart Room

Activated in May 1965, the DSIF chart room (Fig. 2) in the Telecommunications Laboratory, JPL Building 238, is used for the systematic integration of all data generated and used by the DSIF. In establishing such a room, it was intended that the following would be implemented:

(1) A control center for coordination of all engineering and operations information of all DSIF stations.

(2) Effective means and methods of data programming for planning and scheduling DSIF activities.

(3) Communications to provide timely status on all DSIF stations and a system for quick response to scheduling problems for the network.

(4) A focal point for expeditiously transmitting scheduling data when performance, planning, or other requirements have been changed.

The chart room functions primarily as a working area in which the data are received, reviewed, analyzed, and then displayed on either of two types of wall displays. In addition, it serves as a managerial conference area for planning and scheduling.

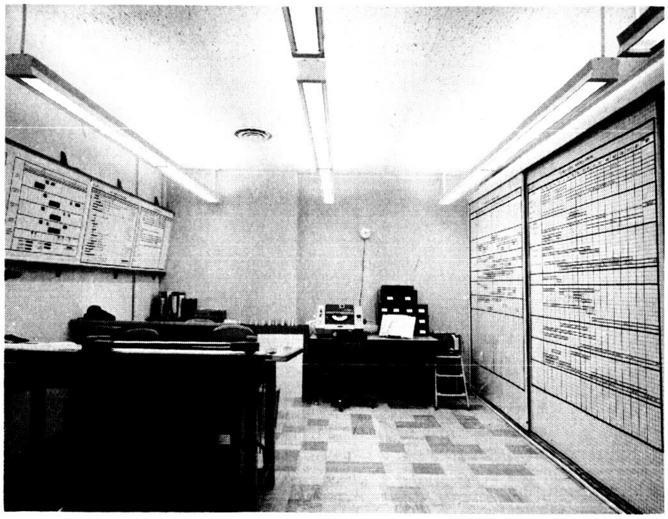

Fig. 2. DSIF chart room

D. GCS Communications Theory Model

A model of the GCS should enable a user to approximate functional relationships within a system so that relationships between input/output transformations and system parameters can be predicted. Such a model is being developed along the following philosophy: It is assumed that the GCS can be represented as a series–parallel network of functional elements. The elements of the general functional block which acts upon input to give output are:

(1) Transformation or transfer function.

(2) Additional noise transformation (for noise with special properties).

(3) Storage capacity, service times, and waiting times.

(4) Information rates or throughput capacity.

The model should: (1) allow the user to quantitatively appraise the bounds on the accuracy of data transferred through a particular system, and (2) provide a basis for the design of codes which will optimally compensate for the system noise within operational constraints.

As part of the GCS model development, a general error model for digital data error is also being developed. This model will be used in conjunction with the GCS model or separately to simulate the effect of a system which introduces error into the digital data. The model will be general enough to be representative of systems which exhibit correlated noise and thus will be of use in the study of the effectiveness of particular coding schemes.

VI. Deep Space Instrumentation Facility

A. Introduction

The Deep Space Instrumentation Facility (DSIF) utilizes large antennas, low-noise phase-lock receiving systems, and high-power transmitters located at stations positioned around the Earth to track, command, and receive data from deep space probes. The DSIF stations are:

Station	Location
Goldstone Pioneer	Barstow, California
Goldstone Echo	Barstow, California
Goldstone Venus (research and development)	Barstow, California
Goldstone Mars (under construction)	Barstow, California
Woomera	Island Lagoon, Australia
Tidbinbilla	Canberra, Australia
Johannesburg	Johannesburg, South Africa
Madrid	Madrid, Spain
Spacecraft Monitoring	Cape Kennedy, Florida
Spacecraft Guidance and Command (under construction)	Ascension Island

JPL operates the U. S. stations, will operate the Spacecraft Guidance and Command Station, and currently plays a major role in the operation of the Madrid Station. The overseas stations are normally staffed and operated by government agencies of the respective countries with the assistance of U. S. support personnel.

The DSIF is equipped with 85-ft-diameter antennas having gains of 53 db at 2300 Mc and a system temperature of 55°K, making it possible to receive significant data rates at distances as far as the planet Mars. To improve the data rate and distance capability, a 210-ft-diameter Advanced Antenna System is under construction at the Goldstone Mars Station, and two additional antennas of this size are planned for installation at overseas stations.

B. Tracking Stations Engineering and Operations

1. Goldstone Pioneer Station

Testing of the *Surveyor* T-21 prototype system-test spacecraft was completed, and the spacecraft was shipped from the Goldstone Pioneer Station. The testing included crew training, system compatibility, and major subsystem tests. Good-quality 200-line pictures were recorded during tests of the spacecraft horizontal camera.

Interface and subsystem tests are continuing, and preparations for DSIF/SFOF (JPL Space Flight Operations Facility) net integration tests are in progress.

The Pioneer Station continued as the overseas staging center for initial testing of S-band equipment and systems. The Spacecraft Guidance and Command Tracking Station equipment was shipped in mid-September, and equipment for the Madrid Tracking Station and the Manned Space Flight Network backup system are currently being assembled.

2. Goldstone Echo Station

The second turn-on of the *Mariner IV* TV equipment was accomplished using the Goldstone Echo Station 10-kw transmitter. From that point until mission termination, the tracking operations were relatively routine. Several experiments were conducted using the spacecraft received signals as reference. A signal-to-noise estimator series of tests was conducted until mission termination. Also, a diversity receiver test was performed using the coordinated facilities of the Goldstone Echo, Pioneer, and Venus Stations.

During the final portion of the *Mariner IV* tracking, only minor equipment difficulties were encountered. Due to a malfunction in the Echo Station maser cryogenic assembly and the resulting necessary maser cooldown time, the Venus Station operated in the receive mode, with the Echo Station recording telemetry for 1 day during the final month. Mission termination was recorded by the Echo Station at 23:05:07 GMT on October 1, 1965.

Installation and testing of the *Pioneer* Project mission-dependent equipment began in April 1965. During the final weeks of the *Mariner IV* mission, interface and operational testing was performed between the end of the station post-tracking calibrations and the start of the next day's tracking countdown. This testing included a series of spacecraft orientations between the Echo Station ground equipment and a *Pioneer* spacecraft test model (Fig. 1). The final interface and integration testing of the S-band system interfaces with the *Pioneer* mission-dependent equipment was begun after the completion of the *Mariner IV* mission. The *Mariner* mission-dependent equipment was removed from the Echo Station, and the station was then readied for a full *Pioneer* Project configuration and for final tests.

Preparations for the arrival of *Lunar Orbiter* Project equipment were begun. With the arrival of major racks of the ground equipment, installation procedures were started. Concurrent with equipment installation, *Lunar Orbiter* computer programs are receiving preliminary operational tests using the S-band digital instrumentation subsystem.

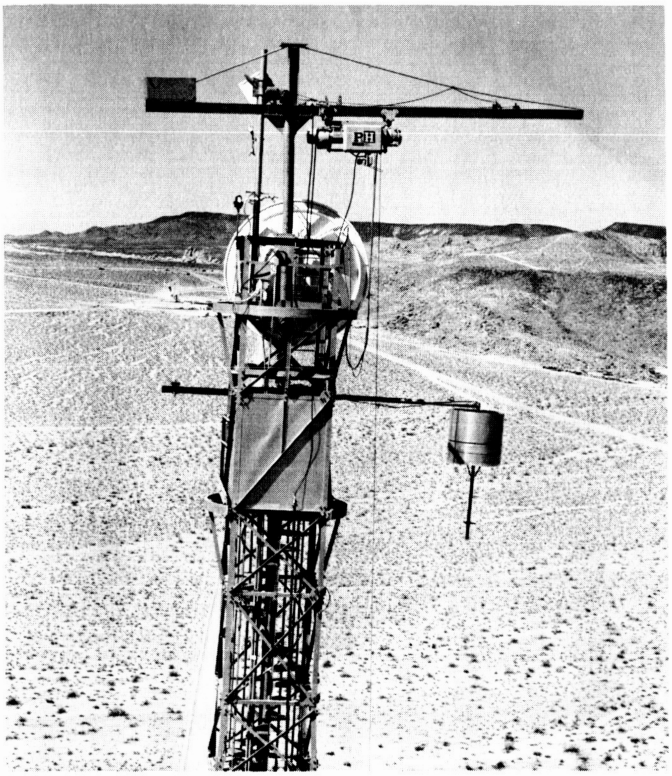

Fig. 1. *Pioneer* test model undergoing spacecraft orientation testing

3. Goldstone Venus Station

Command transmission to the *Mariner IV* spacecraft remained the primary function of the Goldstone Venus Station until mission termination. During the final 2 wk, the Echo Station 10-kw transmitter was able to establish two-way lock with the spacecraft, but had insufficient power to lock up the command loop. When the spacecraft was 192 million miles from the Earth, the full command capabilities of the Venus Station 100-kw transmitter were used. At the time the final command was transmitted on October 1, the round-trip time for a signal to reach the spacecraft and return to Earth was 35 min. This last command caused the spacecraft to switch to the omnidirectional antenna on the spacecraft-to-ground link.

X-Band experiments on the Moon of greater complexity and longer duration than previous experiments were performed at the Venus Station. These experiments included total spectrum, mapping, and open-loop ranging. A measurement of the apparent radio temperature of the Sun at 2295 MHz was made using the 85-ft antenna and a DSIF traveling wave maser. The antenna was boresighted on the Sun, and a comparison was then made between the noise outputs from the receiver when terminated in an ambient temperature load and when terminated in the antenna and looking at the Sun. An apparent solar brightness temperature of 16,400°K at 2295 MHz was indicated. No corrections have yet been made for possible limiting in the maser at the high signal levels represented by this temperature.

4. Goldstone Mars Station

The digital instrumentation subsystem, the station control and monitor console, and the frequency timing subsystems have been installed, and installation operational testing of the digital instrumentation subsystem is under way. Other components of the S-band system are being stored at the Goldstone Echo Station.

C. Developmental and Testing Activities

1. Triode Buffer Amplifier

Since the buffer amplifier for the exciter used to drive the 100-kw *Mariner IV* final amplifier exhibited excessive deterioration in output power, life tests were conducted on a triode amplifier of the type flown on *Mariner IV*. The test results were satisfactory, so two of these amplifiers (one spare) were built with the cavities redesigned to operate in the 2110- to 2120-Mc band. This type of amplifier is temperature-compensated and is normally operated in an air-conditioned Cassegrain cone. Tests on the completed unit over the range from −5 to 125°F showed an output power variation of less than 0.1 db. The only significant temperature effect was a slight variation of the bandwidth.

The new buffer amplifier was installed prior to the scheduled date for sending commands on the low-gain antenna to return the *Mariner IV* spacecraft to a cruise mode. Satisfactory performance of the new amplifier was demonstrated.

2. Lunar and Planetary Radar Module

a. 30-Mc balanced mixer (signal type). The lunar and planetary radar subsystem 30-Mc balanced mixer bandwidth was increased from 20 to 100 kc to support the advanced spectrum analysis experiments. A spare balanced mixer was modified: (1) mechanically, due to physical differences in the crystal filters, and (2) electrically, by increasing the bandwidth of the input stages and matching the filter input and output to optimize passband ripple. To reduce the effect of the mixer output on the bandwidth of the stagger-tuned 455-kc intermediate-frequency amplifier that follows it, the resonant circuit was removed, and a special wideband transformer was installed. Physical restrictions of the module required the development of this special wideband transformer. The bandwidth obtained was further increased by the addition of RF-interference filtering to a value of 71 kc to 1.04 Mc. The modifications of the 30-Mc balanced mixer have been completed, and the unit has been installed in the spectrum receiver at the Goldstone Venus Station with a bandwidth of 100 kc.

b. 30.455-Mc balanced mixer (reference type). A coherent leakage test was performed on a 30.455-Mc balanced mixer, using a laboratory test system approximating the Goldstone Venus Station S-band planetary radar receiver. The coherent leakage level was found to be equivalent to 45 db below threshold. Also, a spectral test of the output of the balanced mixer only was performed. The design goal of a 40-db leakage margin below system threshold was met satisfactorily, establishing leakage compatibility of the new central frequency synthesizer soon to be installed at the Goldstone Venus Station with the S-band planetary radar receiver.

3. Simultaneous Lobing Radiometric Tracking System

The S-band system for the 210-ft-diameter antenna will use a simultaneous lobing angle tracking feed system. A radiometer used in conjunction with the tracking feed will be useful for angle pointing and gain calibrations of the antenna system using radio star sources. This technique is especially important for the 210-ft-diameter antenna, since this antenna does not have a collimation tower.

Testing of the simultaneous lobing radiometric tracking system was recently resumed. The system was installed at the Goldstone Echo Station in conjunction with the 85-ft-diameter antenna and the S-band receiver system. Various radio sources were tracked, and tracking data were taken.

The system was also operated in an experiment designed to eliminate the necessity of a collimation tower when phasing the receiver for automatic tracking of a spacecraft. The simultaneous lobing radiometer and a radio source were used with a signal generator to phase the receiver, and the boresight and phasing were checked against those obtained from the collimation tower. The experiment was performed on the hour-angle axis of the Goldstone Echo Station 85-ft-diameter antenna. The average difference between phase shifter settings obtained from the collimation tower and those from the radio source was approximately 25 deg. The results clearly demonstrate the feasibility of the system; however, further refinement of the technique is necessary before the system can be considered operational.

4. *Automatic Checkout Equipment for Mark I Ranging Subsystem*

The Mark I ranging subsystem will permit precise determination of ranges up to 800,000 km when used in conjunction with an S-band "turnaround" transponder. Developed for the DSIF for tracking lunar probes and monitoring early midcourse maneuvers of planetary spacecraft, it is a digital process control device having many signal interfaces with other subsystems, particularly the S-band receiver/exciter.

To check the operation of the Mark I subsystem completely over an extended period of time, it appeared desirable to provide a means of simulating the operational environment. Special-purpose equipment was designed and built at JPL which utilizes techniques and components of the kind already finding application in the Mark I subsystem. The resulting automatic checkout equipment (ACE), used for JPL acceptance quality assurance at the Mark I subsystem contractor's facility, has proved to be sufficiently effective and inexpensive to warrant the construction of a second unit (recently completed) for use at JPL.

The ACE (Fig. 2) tests the reliability of the Mark I ranging subsystem by repetitively exercising operational modes and monitoring responses to detect erroneous operations. Contained in a single rack, which is approximately 5 ft high, the ACE is composed of four functional assemblies; except for the first, each assembly comprises one block identical to the type used in the construction of the Mark I subsystem. These assemblies are as follows:

(1) Control panel assembly, containing circuitry for: (a) control of the ACE and display of the test status, (b) selection of the reference range number and comparison with the Mark I range number, and (c) simulation of certain functions of the receiver/exciter and data subsystem.

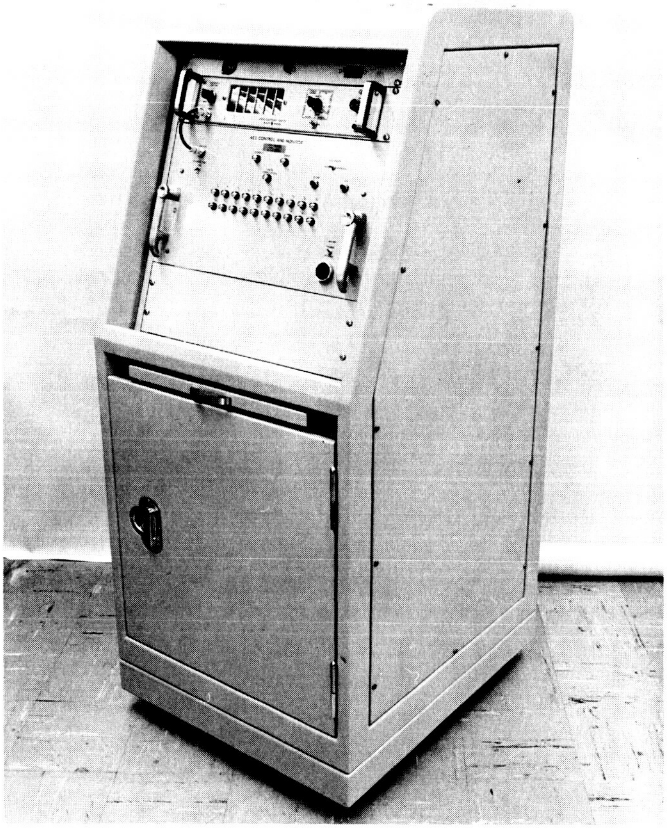

Fig. 2. Automatic checkout equipment for Mark I ranging subsystem

(2) Test states assembly, by means of which the operation of both the ACE and the Mark I subsystem are controlled.

(3) Time generator and counter assembly, which generates the time base which synchronizes the operations within the ACE and provides and controls the doppler signals sent by it to the Mark I subsystem.

(4) Ranging command assembly, which simulates the remaining radio subsystem interlocks, supplies reset and start commands to the Mark I subsystem, and determines the accuracy of the resultant range number.

A test cycle of the ACE with the Mark I subsystem is composed of 20 sequential test states. Of these, the first 15 test the relay interlocks between the Mark I subsystem and the receiver/exciter subsystem. The next four test states test the range code acquisition, with simultaneous input to the Mark I subsystem of doppler information simulating both approaching and receding spacecraft. In the last test state, the resultant range number in the Mark I subsystem is compared with the reference range number in the ACE, which was previously calculated and stored in a binary register.

5. Computer Programs for Mariner IV Occultation Data

As part of the *Mariner IV* Project, three computer programs were written for computing, plotting, and transmitting data related to the occultation of the *Mariner IV* spacecraft. Two of these were written for the digital instrumentation subsystem at the Goldstone Pioneer Station, and one was written for the Scientific Data System (SDS) 930 computer at the Goldstone Echo Station.

The programs written for the Pioneer Station, called the Doppler Predictions Interpolation Program and the Occultation Program, provided for: (1) real-time computation of doppler residuals and integrated doppler phase change from 15 min before to 15 min after *Mariner IV* occultation, and (2) data conversion and teletype transmittal at 100 words/min to the Systems Data Analysis Group at the Pioneer Station and to the JPL Space Flight Operations Facility (SFOF). The program written for the Echo Station, called the 930 Occultation Plot Program, was used by the SDS 930 computer for reading teletype information and plotting the data on the Benson-Lehner point plotter for display by closed-circuit TV at the SFOF.

These three programs may also be used for future spacecraft mission operations.

SUPPORTING ACTIVITIES

VII. Environmental Test Facilities

A. 120-Port Multiple-Pressure-Measuring System

1. Introduction

With the availability of a high-speed data system at JPL, it became worthwhile to design a new multiple-pressure-measuring system (MPMS) for use in wind-tunnel testing and also for pressure measurements in propulsion work. In developing the new system, the following general requirements were met in order to take full advantage of JPL's facilities:

(1) Accuracy of data. The MPMS accuracy should be limited only by the pressure transducers used, which must be capable of operating over a wide range of pressures: 0.004 to 50 psia full scale.

(2) Rate of measurements. Data should be taken as rapidly as possible, not only because of the high cost of wind-tunnel operation, but also to avoid changes in tunnel conditions during data taking. The speed of data taking must be limited only by the settling times of pressures in the tubes from the model.

(3) Range of pressures. Some wind-tunnel pressure-measuring tests involve a greater range of pressures to be measured simultaneously than can be covered by a single transducer. Hence, an array of transducers of various ranges is required. At the same time, it should be possible to disconnect any of the scanners from the system when fewer than the capacity number of pressures are to be measured.

(4) Convenience of operations. A convenient means for servicing scanner units, changing transducers, and making wind-tunnel setups is required.

(5) Range of speeds. The MPMS must be capable of operating at a wide range of speeds in both automatic and manual scanning operations.

(6) Reliability. Reliability is an especially important consideration, since a failure in the data-taking equipment during a test will result in excessive costs to the program.

(7) Portability. Since the MPMS is intended for use with any of several wind tunnels, it must be readily portable.

(8) Use with high-speed data system. The MPMS must be used to full advantage with the high-speed data system.

2. System Description

The MPMS, shown in Fig. 1, has an over-all weight of 150 lb. The 10 sets of knobs visible at the top serve to: (1) connect the inside volumes of the scanners with the "dome" vacuum, (2) disengage the scanners from the drive mechanism, and (3) shift the O-rings for leak testing of the tubing and model.

The 10 12-port scanner units shown in Fig. 1 are arranged in two rows of three units each and one row of four units. The scanner units, with one pressure transducer each, operate simultaneously to feed analog voltages into the data system channels and are read out in a fraction of a millisecond at each port during the scan. Of the 12 ports on each scanner, every first port is connected to a good vacuum, while every twelfth port is connected to an accurately known reference pressure. In this way, accurate zeros and calibrations of all transducers are obtained each time a scan is made. Under these circumstances, 100 ports remain for measuring unknown model pressures.

A variable-speed motor synchronously drives the entire assembly of scanner units through a clutch-brake, a gear box, and a Geneva mechanism. The three shafts and bevel gears through which the scanner units are driven are of negligible backlash. The six-point Geneva mechanism provides intermittent rotation of the rotors. One-third of the total time taken for each port advance is spent in the motion of the advance, and two-thirds is spent with the rotor motionless. Near the end of this interval, the data system makes the readout. During the one-third time in motion, the Geneva mechanism gradually increases the rotational velocity to a maximum and then gradually decreases it to the readout position. This minimizes angular acceleration and deceleration to the near-theoretical minimum, thereby giving smooth operation and minimum wear. Each scanner unit can be disconnected from the drive shaft if it is not needed for a particular test. The clutch-brake provides the possibility, under manual control, of stepping the scanner rotors one port at a time forward and backward. The control is such that the drive shaft always stops with the transducer exactly over a port.

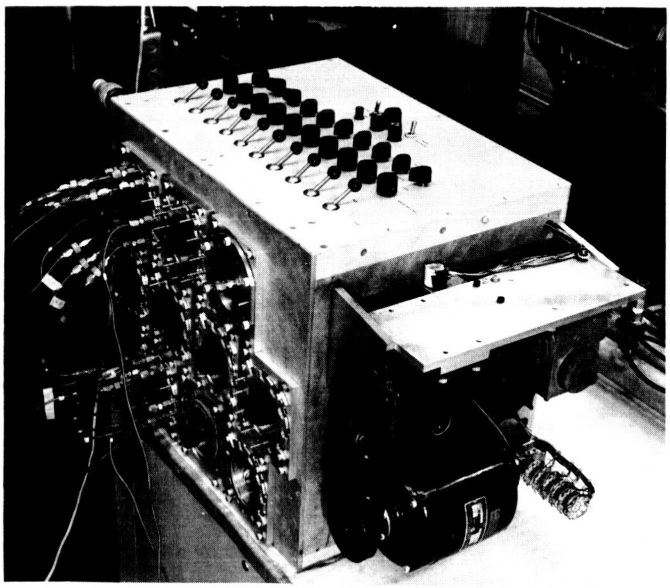

Fig. 1. MPMS assembly

A scanner unit may easily be removed from the assembly. The assembly's design is such that, when a unit is replaced, it is impossible to do so in any way except in synchronism with the other scanner units. Fig. 2(a) shows one of the scanner units removed from the assembly. The 12 plug-in connectors are locked in place by a notched holding ring. The knob makes possible manual rotation of the rotor. The dome cover is made of transparent plastic so that the inside of the unit can be visually inspected. This cover is easily removed for changing transducers and O-rings. A valve in the MPMS assembly for each scanner unit can be shut off before the dome cover is removed in order to prevent loss of dome system vacuum.

A disassembled scanner unit is shown in Fig. 2(b). The holes for the connectors (similar to those used at JPL for several years) are located in the main housing. Underneath these holes are located the added-in volumes. The O-ring retainer permits shifting of the O-rings away from the ports in order to evacuate lines to the model for leak testing. A gap in the retainer permits the O-rings for zero and reference pressures to remain in place during the test so that the zero vacuum and reference pressure systems will not be disturbed.

The transducer in a scanner unit mounts in the rotor and, of course, rotates with it. Since readout always occurs with the rotors stationary over the port being read out, an all-copper slip-ring arrangement on the lower end of the shaft (Fig. 2) can be used for bringing out the transducer leads. Extensive testing has indicated that slip-ring resistances and thermocouple voltages are negligible and that brushes are unaffected by acceleration and will last for years. Since there are no flexible transducer cables in the units to wind up, the rotors can be rotated in either direction as many revolutions as desired. Also, since the transducers are located in the rotor immediately over the rotor port, the associated volume is almost negligible.

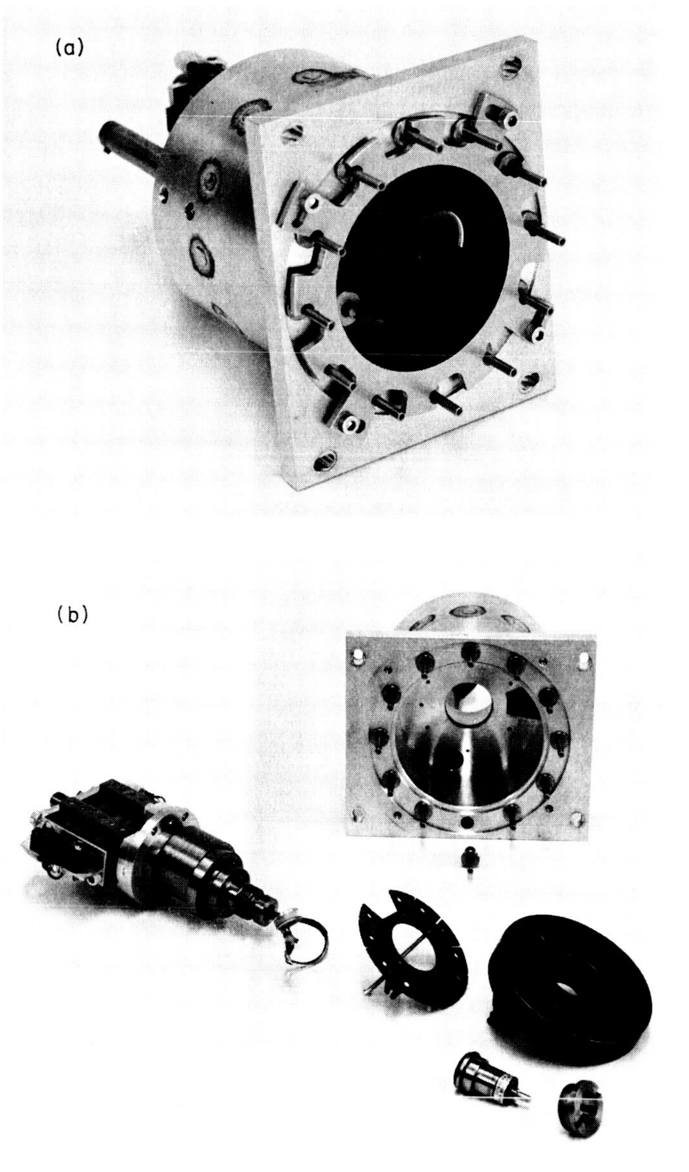

Fig. 2. MPMS scanner unit

Measurements have shown the settling time resulting from the transducer getting connected to a port (connection transient) to be less than 50 msec at a pressure of 0.01 psia. This settling time is independent of the settling time associated with the long tubing between the MPMS and the model.

The added-in volumes slightly increase the settling times of the pressures in the tubes to the model, but, as described below, effectively speed up operation for low-pressure measurements and do not affect high-pressure measurements. When wind-tunnel measurements are being made, it is necessary to wait until the various pressures to be measured have simultaneously settled in all tubes to the model. (These tubes should be as short as possible.) For a pressure of 0.02 psia or less, this can be as long as 5 min. During a scan, at the instant the transducer connects to each port, the associated transducer volume (even though small) causes a second disturbance in addition to the connection transient to the settled pressure in the line of this port. As an example, with no added-in volumes, a 0.048-in.-ID, 16-ft-long tube to the model causes a pressure disturbance of 2 or 3% at 0.01 psia which persists for a time very much longer than the 50 msec of the connection transient. However, for lines with added-in volumes of 1 cm^3, which is 100 times the associated transducer volume, the disturbance in each line is reduced to less than 1%. If desired, this small error can be corrected in the data reduction, thus effectively eliminating it. After settling is complete, the scan can be made as rapidly as desired. The mechanical design of the MPMS is such that a speed of 4 ports/sec maximum can be attained, at which speed the 12 ports of each scanner unit will be run through in 3 sec; i.e., 120 ports are read out in 3 sec. There is no minimum speed.

The scanner units are arranged for rotating 10 of the 12 O-rings one-half port distance by means of a knob (Fig. 1). With O-rings rotated or shifted, dome vacuum is applied to all the lines to the model. With the pressure holes on the model plugged, the vacuum is held after the return shift by those tubes which have no leaks. A scan made immediately will identify leaks as indicated by the tubes with pressure increases.

A microswitch located on the drive shaft just ahead of the Geneva mechanism gives the signal to the data system at the proper time for read-out for each port position. At this instant, all 10 transducers are read out within a fraction of a millisecond. A second microswitch indicates end-of-scan and stops the automatic advancing. A 12-position switch for giving port positions provides numbers as contact closures 01, 02, 03, · · · , 12, which are fed into the high-speed data system.

3. *Vacuum Cart for Operating Pressures*

Fig. 3 shows the MPMS on the vacuum cart table. The DC-200 oil micromanometer is used for calibrating the 10 transducers at pressures of 0.2 psia and less. The vacuum cart supplies the dome, reference, and zero (vacuum) pressures needed for operating the MPMS. Fig. 4 shows each of the three pumping systems. The reference system has a tank from which air can be pumped out, or bled in, to provide a constant pressure for calibrating transducers. The zero-pressure system has a diffusion pump to provide the very good vacuum which is applied to the transducers

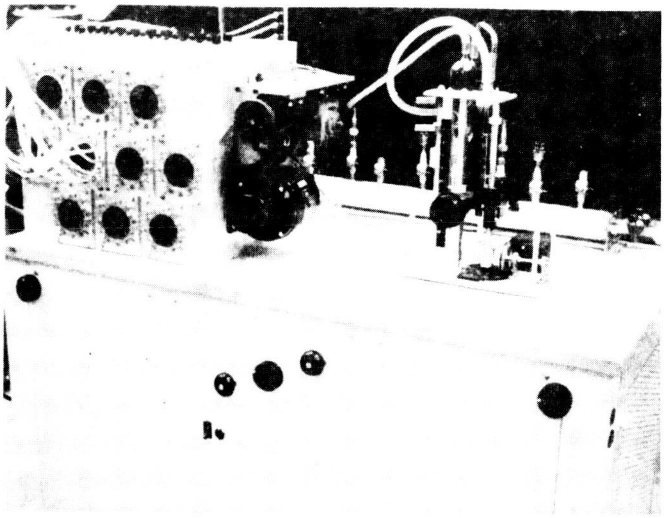

Fig. 3. MPMS on vacuum cart table

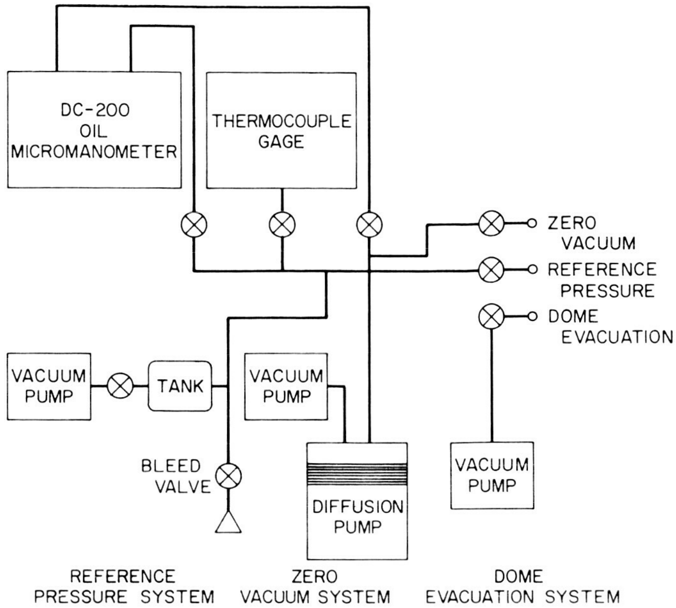

Fig. 4. Vacuum cart schematic

for obtaining zero readings. The reference pressure above 0.2 psia up to 1 atm is measured by a mercury manometer. For pressures above atmospheric, a different system with suitable gages is used.

4. System Performance

The accuracy, limited only by the transducers, is generally about 0.25% for pressures above 0.2 psia. Below this level, the accuracy decreases, becoming about 1% at 0.02 psia.

The fastest scanning time for the 120 ports is 3 sec after the pressures in the tubes have settled. The system can be run slower if desired. With the 10 scanner units built into the MPMS, transducers with 10 different ranges can be used for simultaneously measuring a pressure distribution ranging from 0.001 to 50 psia. With strengthened structures, still higher pressures can be simultaneously measured. With Teflon-coated rotors, the MPMS measures pressures up to 600 psia. The transducer in a scanner unit can be changed in about 5 min, and it is not necessary to shut down the dome vacuum system during the operation. The 14- × 18- × 30-in. MPMS is readily portable and has a very high mechanical reliability.

SPACE SCIENCES

VIII. Space Instruments

A. Tests of the *Mariner C* TV Shutter Solenoid in a Space Molecular Sink Simulator

Introduction. When it was decided to initiate an early encounter science turn-on during the *Mariner IV* Mars encounter, it became necessary to obtain additional data on the life expectancy of the television camera shutter in the space environment. The pacing item affecting the operational life of the shutter is the rotary solenoid. Because of this, additional life testing of the solenoids was proposed.

In order to start the mission with a relatively "fresh" solenoid, new solenoids were retrofitted on all of the flight shutters just before the final spacecraft system tests at the Air Force Eastern Test Range. To successfully complete the *Mariner IV* photographic mission, the life requirement on the solenoid was from 2600 to 2800 exposures, which includes the exposures put on the shutter during subsystem and spacecraft system tests after the solenoid retrofit and during the February 1965 encounter science turn-on and the exposures necessary to complete the mission during Mars encounter. The operational life of solenoids tested at ambient conditions ranged from 16,000 to 29,000 exposures, but no data were available on the life expectancy of the solenoids in the clean, hard vacuum of space.

To realistically simulate the lubrication and mechanical wear conditions existing in the solenoid in the space environment, an extremely clean, ultrahigh-vacuum test environment was required. No test chamber at JPL could provide such a test environment. However, a system called a space molecular sink simulator (molsink), developed by Section 375 of the Jet Propulsion Laboratory under the Office of Advanced Research and Technology Program No. 124-09-04-01 for experimental research, which could provide a clean, ultrahigh-vacuum test environment was made available for life-testing additional solenoids.

From these tests it was hoped that a more realistic determination of the shutter life expectancy could be made.

The solenoid. The rotary solenoid used to actuate the *Mariner IV* shutter was a stock commercial solenoid modified to meet JPL's requirements for flight hardware. The rotary motion was produced by three hardened steel balls rolling in inclined ball races as the armature was pulled down by the magnetic force of the coil. The total rotary motion of the solenoid was 45 deg. A coil spring attached to the armature shaft and the solenoid base produced the return motion when the coil current was turned off. Fig. 1 is an exploded view of the solenoid.

The solenoid was lubricated with a solid film lubricant (MoS_2 in a sodium silicate binder), which was sprayed onto the surfaces and then heat-cured. Its primary function in the solenoid was to prevent metal-to-metal contact between moving surfaces, although it had some lubricating properties. Metal-to-metal contact was avoided because of the possibility of cold welding. The solid lubricant, of course, eventually wears away by the abrasive action of the moving surfaces. This abrasive action was particularly severe between the hardened balls and the ball races because of the extremely high contact stresses (100,000–200,000 psi). The limited wear life of this solid film lubricant determined the useful life of the solenoid.

The *Mariner C* TV shutter required two solenoid cycles to complete one shutter exposure. That is, the first solenoid cycle opened the shutter and the second cycle closed the shutter. The exposure time was the time between the first and second solenoid pulses or cycles.

Vacuum chamber description. The space molecular sink simulator consists of an 18-in. diameter by 30-in. Varian stainless-steel bell-jar vacuum system which uses

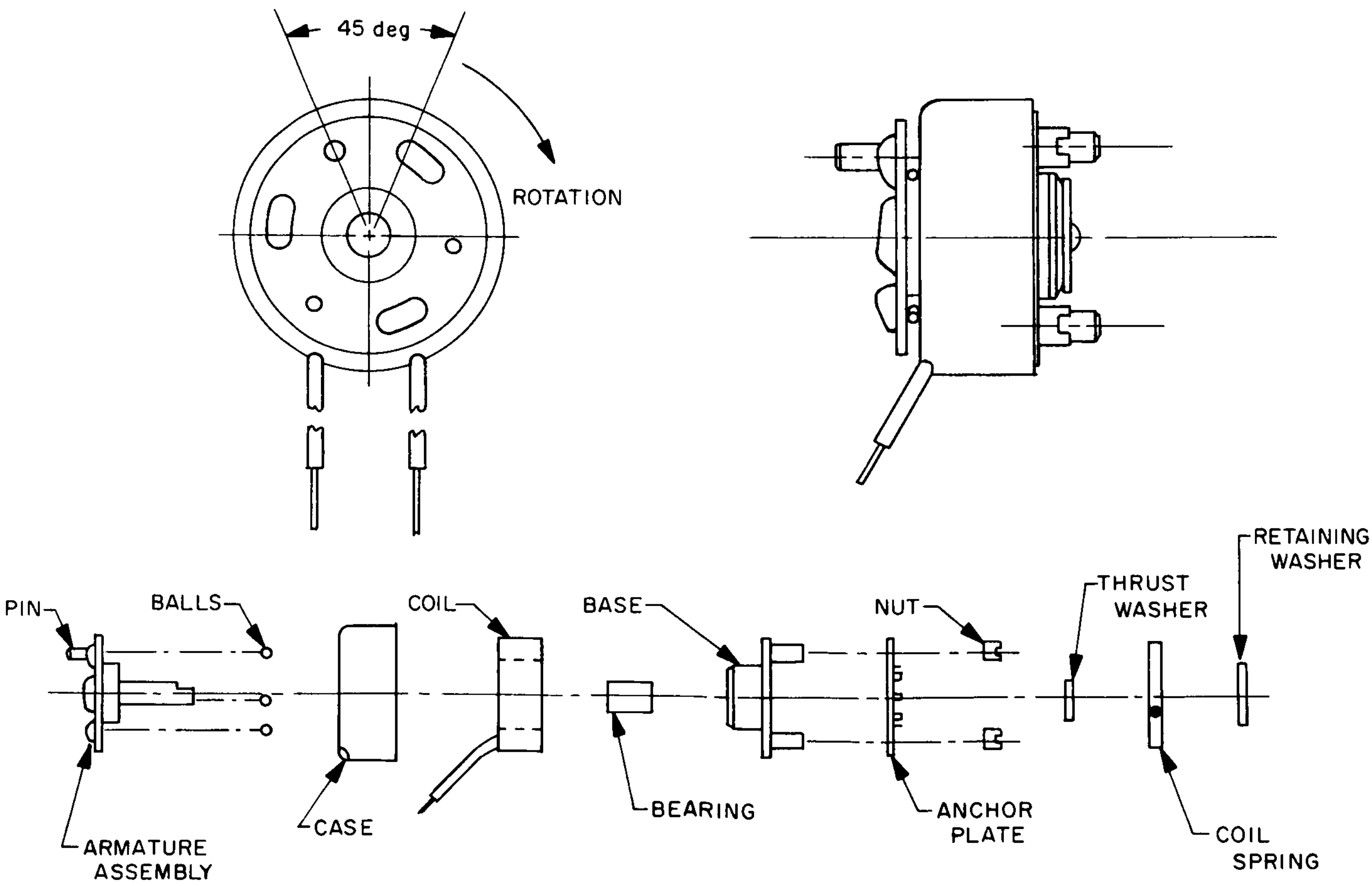

Fig. 1. Rotary solenoid, *Mariner* C TV shutter (exploded view)

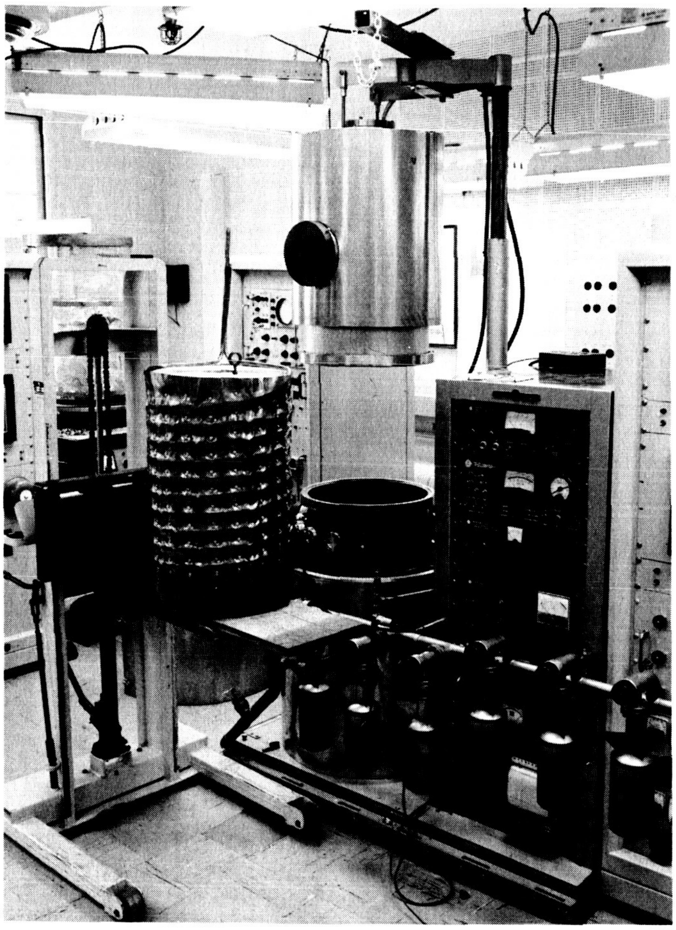

Fig. 2. Space molecular sink simulator

all metal seals and is ion-getter and titanium sublimation pumped. Fig. 2 depicts the chamber with the LN_2-cooled molecular trap removed. Fig. 3 is a view of the inside of the molecular trap array. The longitudinal sharp wedge fins of the array provide an order-of-magnitude capture improvement for condensible molecules emanating from the test item over that provided by a smooth wall at the same temperature. This arrangement will capture all but one out of every 10,000 condensible molecules (H_2O, oil, plasticizers) emanating from the test item before they can restrike it. Since the test item is the major source of gas within the molecular trap, this ratio and the actual outgassing rate of the test item determine the monolayer formation time at the surface of the test item.

A very rough estimate of the monolayer formation time can be made by estimating the test item gas evolution rate and also the capture coefficients of these gases upon the molecular trap surfaces and upon the test item surfaces. Combining these estimates with the known dimensions and configuration of the molecular trap and the test item, it could be assumed that the monolayer formation time was long with respect to the cycle time of the solenoids tested. For example, if the solenoid released one monolayer of water per second (2×10^{-5} torr-liters/cm^2/sec) from its exposed surfaces, it would take 1 hr for a monolayer of water to form on a cleaned (bearing surface) surface, provided all of the molecules that struck that surface stuck. The indigenous noncondensible gas pressure (nitrogen was the only gas detected by the residual gas analyzer) was measured to be about 1×10^{-9} torr. This provided a monolayer formation time of about 5 hr if it were assumed that only 10% of the nitrogen that struck the clean surface stuck. Further explanation of the space molecular sink chamber and its operation can be found in *SPS 37-30* and *37-34*, Vol. IV.

Test description and procedure. A total of nine solenoids was life-tested in the molecular sink simulator during the period of June 15–July 21, 1965. Five of the solenoids were given an initial run-in of 1400 to 1800 exposures at ambient temperature and pressure before placing them in the molsink chamber. The other four solenoids were not given the 1400 to 1800 exposure run-in before the test, but were turned on only long enough to verify that they were operating properly.

The reason for the initial run-in of five of the solenoids was to find out if a run-in of 1000 to 2000 exposures at ambient conditions had any effect on the operating life of the solenoid in a high vacuum. The solenoid on *Mariner IV* had 1500 exposures put on it at ambient conditions before launch.

The solenoids were mounted on a test fixture similar to the one shown in Fig. 4. Several fixtures were fabricated to accommodate the various numbers of solenoids that were tested in the chamber. The temperature of the solenoids was controlled by a heater wire soldered to the solenoid mounting plate and was monitored by a thermocouple attached to the mounting plate or the solenoid itself. Power leads from the solenoid were soldered to vacuum-sealed terminals on the top plate of the fixture. The top plate, which serves as a vacuum seal, was mounted on the top of the chamber with the solenoid mounting plate suspended from it inside the chamber.

After the solenoid was mounted in the chamber, it was operated a few times to verify that it was working properly. The chamber was then pumped down to the 10^{-8}-torr range and the system baked out at 175°F for 48 hr. The bake-out cleaned the system of condensible molecules

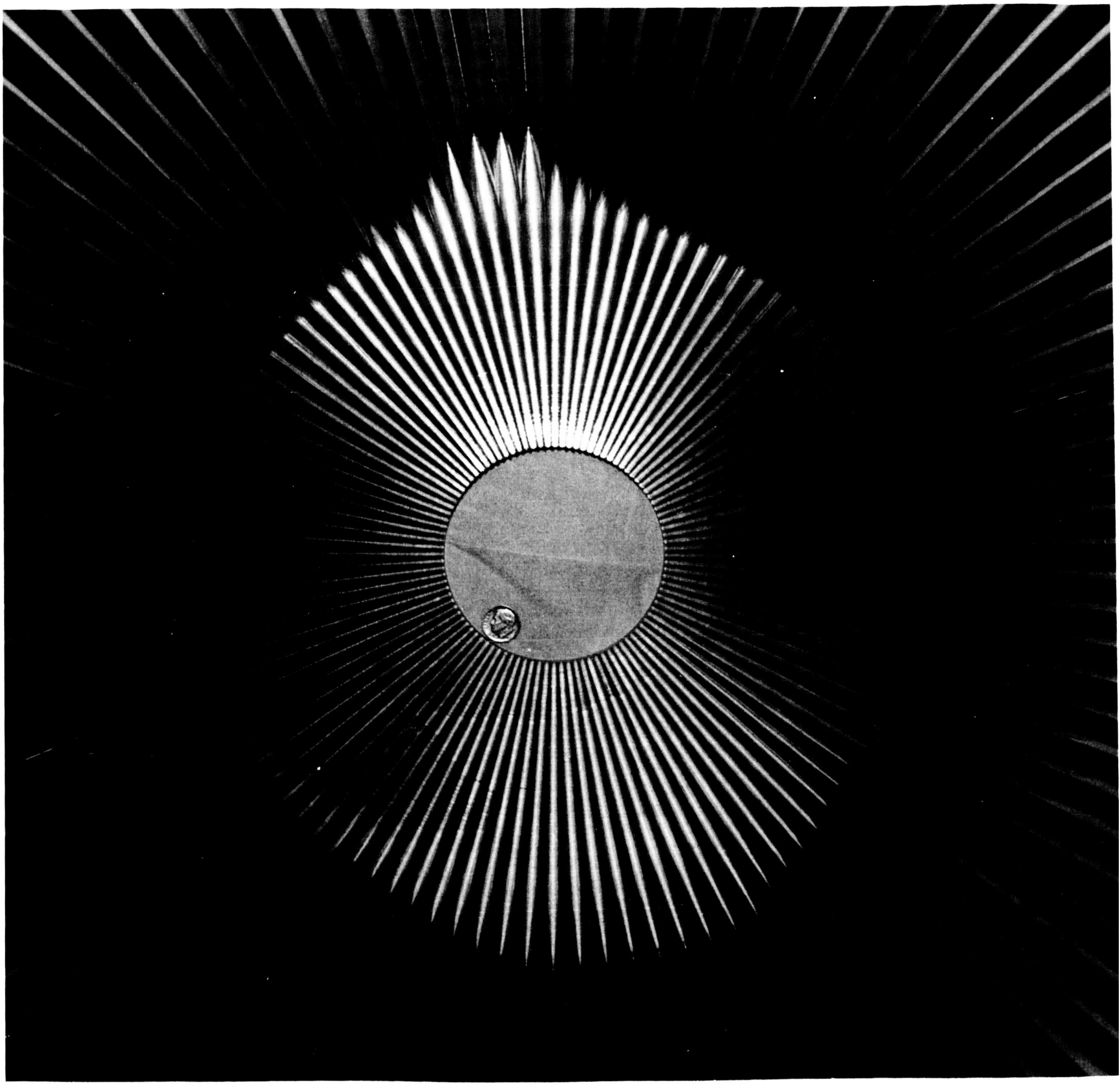

Fig. 3. View of the inside of the molecular trap

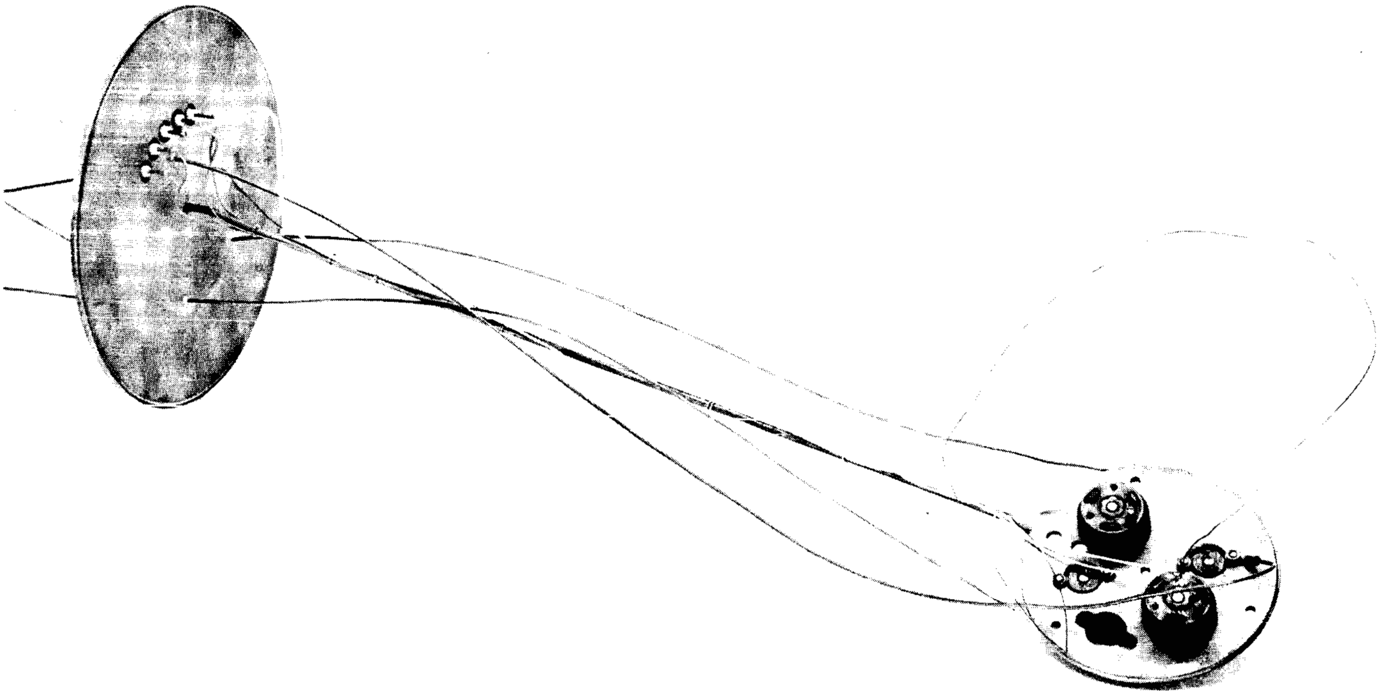

Fig. 4. Solenoid mounting fixture

and provided an accelerated vacuum soak for the solenoid, which helped simulate the effect of the 7½-mo exposure of the flight solenoid to space. The 175°F bakeout temperature did not exceed the maximum rated temperature of the solenoid, which was 257°F. All of the flight solenoid coils had been outgassed at 320°F in an argon atmosphere for 1 hr, and the assembled solenoids at 230°F for 22 hr in a vacuum. At the end of the bakeout period, the chamber cold wall was turned on and cooled down to −210°F. The solenoid temperature was maintained at +14°F by the heater wire. When the cold wall and solenoid had stabilized at their respective temperatures, the solenoid was turned on and the test started.

Two shutter pulsers, which were built for testing the flight shutters, were used to operate the solenoids during the tests. The pulsers used the same shutter drive circuit that was used in the TV camera head. The flight shutter operated at the rate of one exposure every 48 sec. Due to testing time limitations, the solenoids were operated during the tests at an accelerated rate of one exposure per 5-sec interval. An exposure was two solenoid pulses of 25-msec duration spaced 200 msec apart.

Solenoid operation during the tests was monitored by displaying the coil current on an oscilloscope and periodically observing and photographing the current waveform. Typical current waveforms are shown in Fig. 5. The shape of the waveform was affected by the moving armature, due to the back emf induced in the coil. Basically, three things were determined from the shape of the waveform:

(1) That the solenoid moved when pulsed.

(2) How fast it moved.

(3) Amount of ringing or bounce at the end of the power stroke.

The amount of ringing of the armature at the end of the power stroke indicated the amount of friction in the solenoid. That is, a decrease in the amount of ringing indicated an increase in friction. The solenoids were also monitored by listening to their operation, which could be heard outside the chamber. In some tests the solenoids were operated all night. In this situation they were monitored by recording the signal from an accelerometer, which was mounted on the chamber top on a strip recorder. This produced a spike on the strip chart when the solenoid was pulsed.

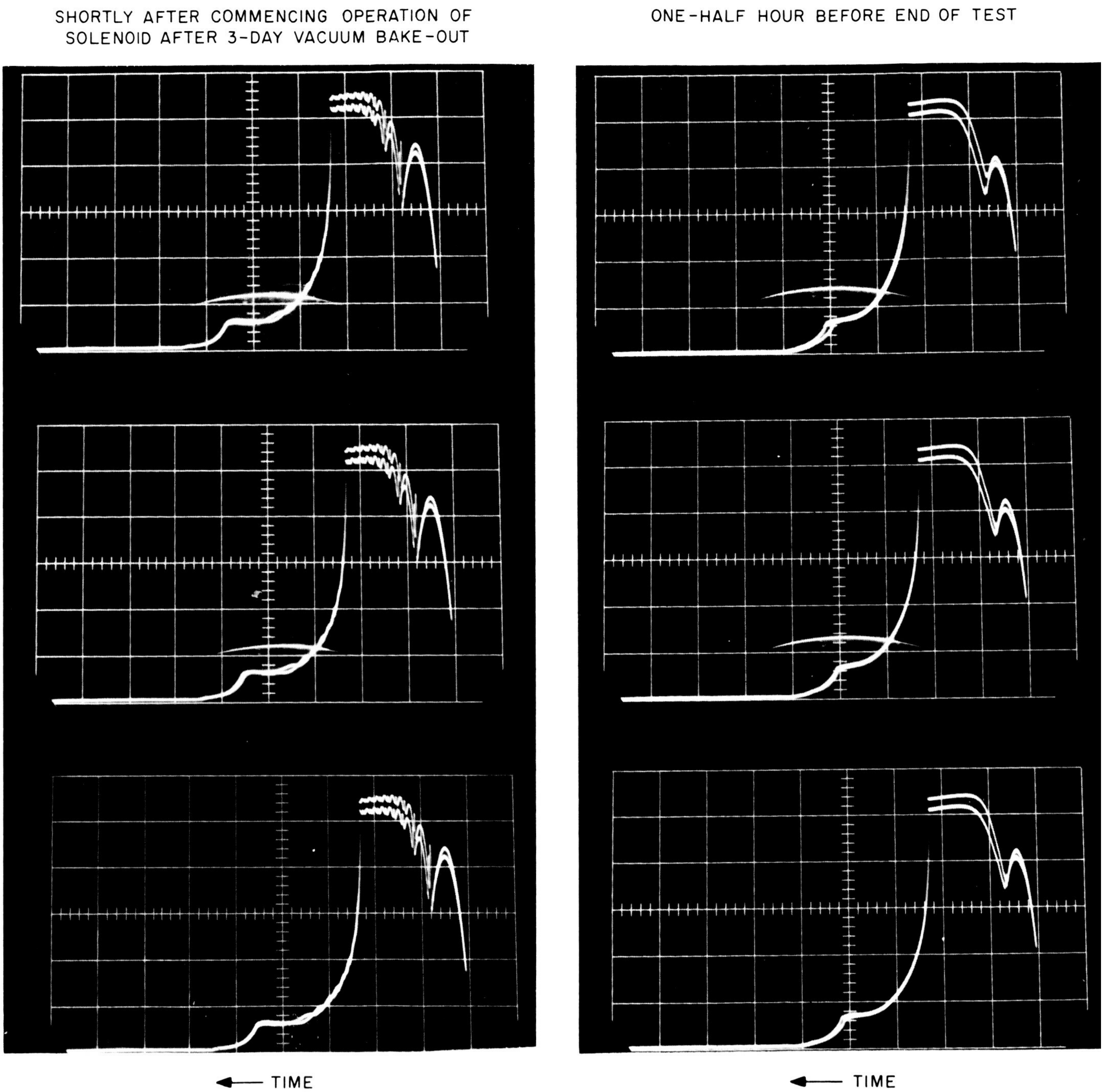

Fig. 5. Solenoid No. 204 coil current during JPL molecular sink testing

Solenoid life test results. The results of the solenoid life tests are given in Table 1. The loaded test condition means that the solenoid was tested while assembled in a *Mariner C* TV shutter. The number of exposures to failure given is the number put on a solenoid during the test. It does not include the pretest run-in exposures. The initial failure occurred when the solenoid first failed to actuate on a pulse. The complete failure occurred when the sol noid failed for several minutes to actuate when pulsed.

Conclusions. The tests showed that the solenoi had a considerably shorter life expectancy operating the molsink than they did operating in air. The opera ing life of the solenoids tested in air ranged from 16,000

Table 1. ***Mariner*** **C TV shutter solenoid life test results**

Test Type	Test No.	Solenoid Serial No.	Date of test	Solenoid test conditions: Temperature, °F	Solenoid test conditions: Pressure, torr	Solenoid test conditions: Solenoid loading	Number of exposures: Before test	Number of exposures: To initial failure	Number of exposures: To complete failure	Remarks
Bench test		217	9/23/64 to 12/2/64	+75	Ambient	Unloaded	Unused	26,000		The bench tests at ambient conditions were conducted in the Space Optics Group Laboratory. The test of each solenoid was ended after the onset of the initial failure mode.
		223				Unloaded	Slightly used	19,000		
		205				Loaded	Unused	16,000		
		213				Loaded	Unused	29,000		
Molsink tests	1	207	6/15/65	+14	4×10^{-8} to 3×10^{-9}	Unloaded	Unused	7,620	9,060	Microscope examination of this solenoid showed evidence of a misaligned armature, which accelerated the abrasion of the solid lubricant in one ball race.
	2	204	6/18/65 to 6/24/65	+14	5×10^{-9} to 3×10^{-10}	Unloaded	1,775		15,000	These two solenoids were mounted on one fixture and operated simultaneously during the test by two shutter pulsers. Solenoid 204 did not fail; the test plan was limited to 15,000 exposures.
		212				Unloaded	Unused	4,000	7,900	
	3	215	6/25/65 to 6/30/65	+14	5×10^{-9} to 3×10^{-10}	Unloaded	1,400	8,340	8,820	The solenoids in this test were mounted and operated in the same manner as in test No. 2.
		221				Unloaded	Unused	11,130	13,890	
	4	201	7/2/65 to 7/7/65	+14	4×10^{-9} to 8×10^{-10}	Loaded	Unused	14,715	17,415	
	5	219	7/8/65 to 7/21/65	+37	3×10^{-9} to 6×10^{-10}	Unloaded	1,440	7,260	19,485	In this test all three of the solenoids were mounted on one fixture in the chamber, but only one solenoid was operated at a time.
		220				Unloaded	1,440	14,340	No data	
		224				Unloaded	1,440	10,920	11,040	

29,000 exposures, while the life of those tested in the molsink ranged from 4,000 to 15,000 exposures.

The type of failure that occurred in the solenoid was a mechanical lock-up between the steel balls and the ball races, which prevented the solenoid from actuating when pulsed. This lock-up was preceded by a decrease in the armature bounce (ringing at the end of the power stroke) and a slowing of the armature movement.

Microscopic examination of three of the solenoids tested to failure in the molsink showed that the solid lubricant had been completely removed from the bottom of the ball races by the abrasive action of the steel balls. It was this complete removal of the solid lubricant, producing metal-to-metal contact, that was considered to be the cause of the solenoid failure. The solid lubricant appeared to provide adequate lubrication of the solenoids in a high vacuum. There was no difference in operating characteristics of the solenoids when operating in air and when operating in the molsink at the start of the test.

Although the high vacuum in the molsink had no apparent effect on the operating characteristics of any of the solenoids at the start of the test, the lack of absorbed molecules between friction surfaces probably accelerated the abrasion of the solid lubricant from the bearing surfaces. It was speculated that, after the bare metal surfaces started to come in contact with one another, a form of microscopic cold welding took place during the vacuum test. This speculation was supported by the fact that, when the chamber was back filled with a dry gas, the solenoids then proceeded to operate again. The dry gas may have acted as a separating wedge on the microscopic cold welds. In the case of bench test (ambient conditions) failure, it was presumed that the wear proceeded until sufficient galling occurred to produce a lock-up condition which could not be broken by the pulsing force of the solenoid.